

COMPARISON OF SOLAR ELECTRIC AND CHEMICAL PROPULSION MISSIONS

Joshua E. Freeh

NASA Glenn Research Center, USA, joshua.e.freeh@nasa.gov

Laura M. Burke

NASA Glenn Research Center, USA, laura.m.burke@nasa.gov

Waldy K. Sjauw

NASA Glenn Research Center, USA, waldy.k.sjauw@nasa.gov

Melissa L. McGuire

NASA Glenn Research Center, USA, melissa.l.mcguire@nasa.gov

Bryan K. Smith

NASA Glenn Research Center, USA, bryan.k.smith@nasa.gov

Solar Electric Propulsion (SEP) offers fuel efficiency and mission robustness for spacecraft. The combination of solar power and electric propulsion engines is currently used for missions ranging from geostationary stationkeeping to deep space science because of these benefits. Both solar power and electric propulsion technologies have progressed to the point where higher electric power systems can be considered, making substantial cargo missions and potentially human missions viable. This paper evaluates and compares representative lunar, Mars, and Sun-Earth Langrangian point missions using SEP and chemical propulsion subsystems. The potential benefits and limitations are discussed along with technology gaps that need to be resolved for such missions to become possible. The connection to NASA's human architecture and technology development efforts will be discussed.

I. INTRODUCTION

NASA is developing human missions and architectures to Mars.¹ While the success of nearly 15 consecutive inhabited years on the International Space Station is significant, missions to Mars require additional progress in many spacecraft subsystems and other areas to successfully complete these missions.

To mitigate existing challenges, NASA is evaluating space missions between low Earth orbit and the Mars surface. As these missions get further from Earth orbit, everything from communication to mission abort scenarios becomes more challenging and more Mars-like. They provide stepping stones to the goals of humans on the surface of Mars. These missions can also be valuable from both the scientific and exploration perspectives too, along with the technology development.

Solar electric propulsion is one combination of technologies that can be valuable for Mars missions. Solar power systems continue to become more capable in space, with better conversion efficiency, lower mass, and higher power due to larger scale designs and packaging.² Electric propulsion systems also continue to

improve as fuel efficient in-space thrusters, by extending their design life, operating at higher power, and better efficiency.³ The combination of these technologies provides a highly capable power and propulsion capability for a variety of different missions from geostationary satellites⁴ to interplanetary science missions.⁵ New thrusters and solar arrays⁶ have enabled other missions including both cargo and potentially crew (in hybrid SEP/chemical combinations) to Mars.⁷ NASA has been developing and funding both of these important technologies because of their potential widespread impact for space transportation.

This paper compares SEP and chemical propulsion systems for three different cargo missions: a mission to the moon, Mars, and the James Webb Space Telescope Sun-Earth L2 Lagrangian point orbit.⁸ These missions offer a variety of distances and scenarios to evaluate the two systems and are each relevant to the proving ground and Mars missions that NASA is developing.

II. MODEL AND ASSUMPTIONS

II.I. SEP Spacecraft Description and Assumptions

The solar electric propulsion version of the model is based on a notional NASA SEP technology demonstration mission spacecraft. One demonstration mission being evaluated is the Asteroid Redirect Mission (ARM).⁹ For this mission, a robotic spacecraft would rendezvous with a large near-Earth asteroid, grab a boulder from its surface, and transport it to a stable lunar orbit. A crewed mission would then rendezvous with the robotic spacecraft to sample the boulder and return to Earth.

This paper uses this notional SEP spacecraft as a cargo (non-crew) payload transport. The amount of payload capability will vary with the mission but be constant for the different propulsion systems.

The solar power subsystem has been sized to provide 45 kW electrical power at the end-of-life and at 1 AU distance from the sun. The electrical power is assumed to change with distance from the sun proportional to $1/r^2$, where r is the distance from the sun in AU. The spacecraft is assumed to require 5 kW electrical power at all times, so the minimum power available to the electric propulsion system is 40 kW. The solar arrays would then be sized appropriately to meet the 45 kW end-of-life power requirement based on the radiation dose and other considerations throughout the mission. Any extra electrical power greater than the end-of-life power is not used for the thrusters.

The electric propulsion subsystem is assumed to have three Hall effect thrusters plus one spare thruster.¹⁰ Each thruster string would have approximately 13.3 kW of available electrical power from the solar array. Xenon is assumed to be the propellant and for this study, the maximum xenon storage required is 5000 kg. Each thruster is assumed to have a specific impulse between 2000 – 3000 seconds, depending on power level and thrust requirements. The thrusters are specified to provide thrust for a maximum of 90% of the mission to account for periods of no thrust throughout the trajectory.

The notional spacecraft dry mass, without payload, is estimated to be 3500 kg. This includes the power and propulsion subsystems described above along with structural, thermal, communication, guidance, navigation, and control, launch and payload interfaces, and other subsystems. In later stages of its design, the dry mass could vary with xenon storage and payload mass, but for this comparison, a constant dry mass (without payload) is sufficient. Another scenario is that this cargo spacecraft could be a second build of the ARM robotic spacecraft, in which the same design and

drawings are used to build a serial number two. In this case, the xenon storage could vary as needed for the mission, while the spacecraft structure, xenon tanks, and other components would remain fixed for the maximum xenon storage and payload capability.

II.II. Chemical Propulsion Spacecraft Assumptions

The chemical propulsion version of the spacecraft uses the same dry mass without payload of 3500 kg as the SEP case. While there would certainly be differences in propellant storage and propulsion system design, this approach allows a first order comparison of these systems. The propulsion model assumes simplified impulsive orbital transfers and a specific impulse of 300 seconds.

For both propulsion cases, an optimized spacecraft would require iterations on spacecraft design, mass, trajectory, and propellant use.

II.III. Trajectory Assumptions

The trajectories have been chosen to clarify and isolate the comparison of the two propulsion approaches as much as possible. Each trajectory requires additional propulsion before the start points and after the end points.

For the Earth orbit to lunar orbit cases, the trajectory begins at the nominal International Space Station low Earth orbit and finishes at a lunar distant retrograde orbit (DRO) at approximately 60,000 km altitude.

For the Earth orbit to James Webb Space Telescope orbit cases, the trajectory begins and ends at the Earth-Moon L2 Lagrangian point. The trajectory enters the James Webb Space Telescope orbit, which is a Lissajous orbit around the Sun-Earth L2 Lagrangian point.

For the Earth escape to Mars entry cases, the trajectory is between Earth escape and Mars entry, where the characteristic energy, C_3 , equals 0 km^2/s^2 in both locations. From the Mars entry to lower Mars orbit or Mars surface, chemical propulsion, SEP, and/or aeronautic drag could be used.

II.IV. Software

COPERNICUS¹¹ was used to model the majority of the trajectories shown here. It is a three degree-of-freedom spacecraft trajectory design and optimization model.

CHEBYTOP¹² was used to model the Mars SEP case. It is a general-purpose two-body, sun-centered, low thrust trajectory optimization software tool that is ideal for this case.

III. RESULTS AND DISCUSSION

III.I. Earth Orbit to Lunar Orbit

The SEP trajectory, shown in Figure 1, is a long duration spiral transfer that is characteristic of a low thrust SEP orbit change.

Two chemical propulsion trajectories are included, Figures 2 and 3. Both use a lunar flyby; the latter takes advantage of the Weak Stability Boundary (WSB). The WSB trajectory has a longer travel time but requires less propellant.

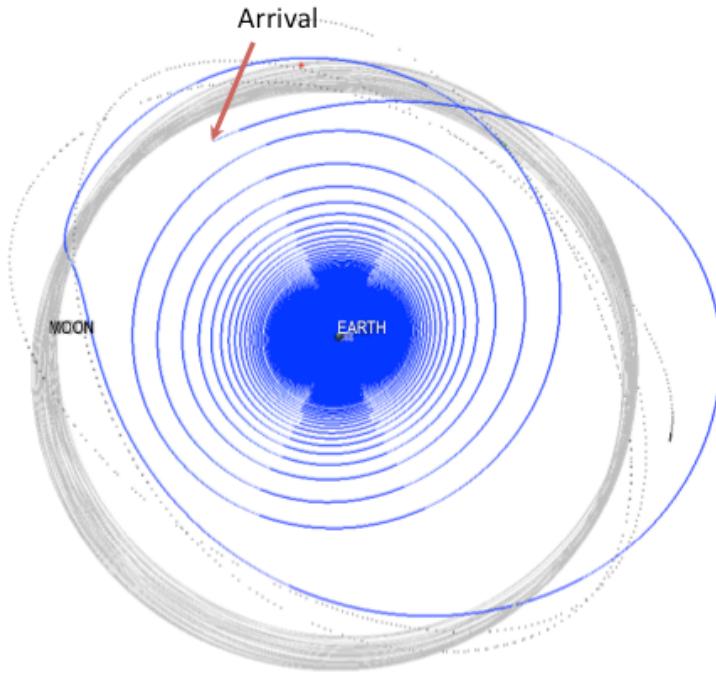


Figure 1: Plot of the SEP spacecraft trajectory from low Earth orbit to lunar distant retrograde orbit. The trajectory is a spiral transfer with many orbits of the Earth.

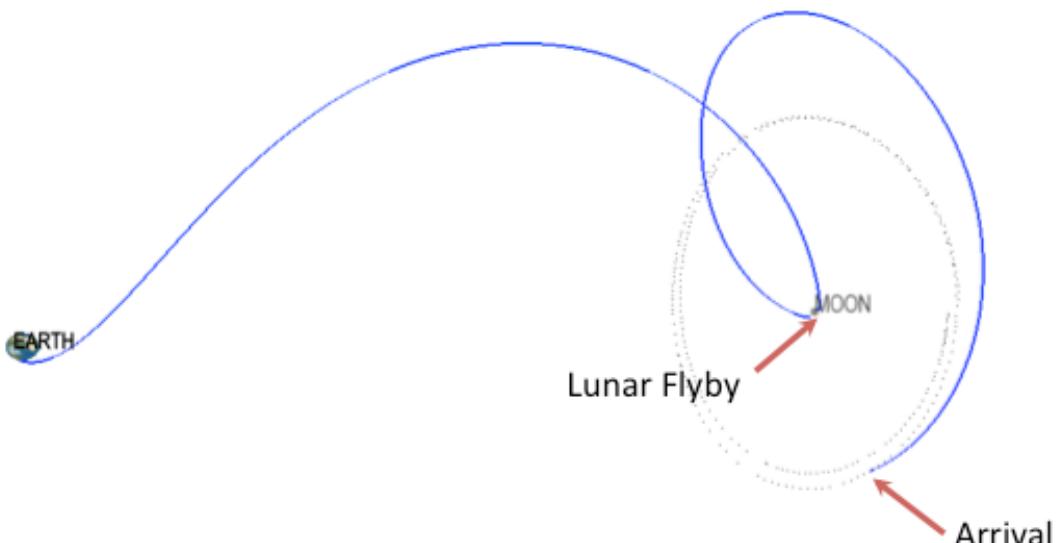


Figure 2: Plot of a chemical propulsion spacecraft trajectory from low Earth orbit to lunar distant retrograde orbit with a lunar flyby.

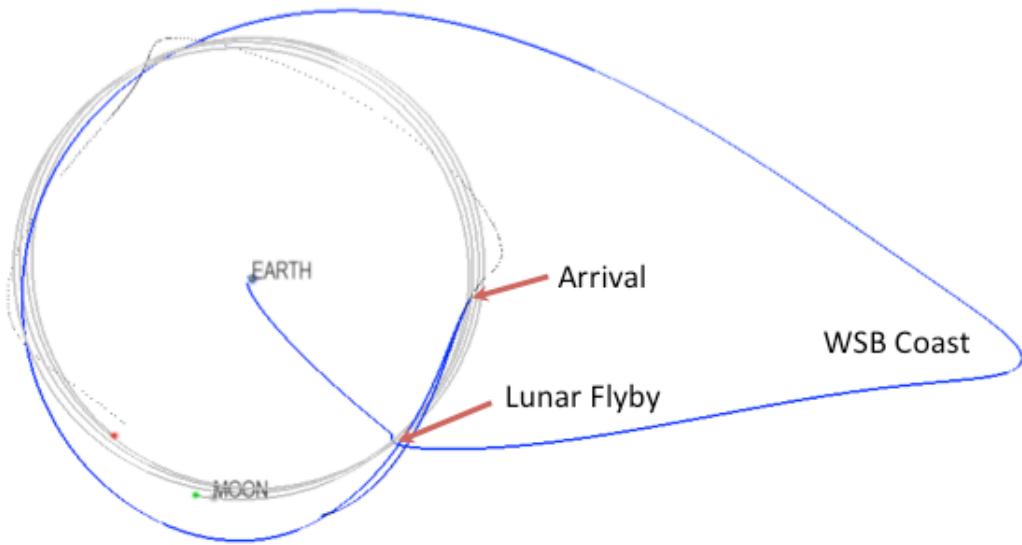


Figure 3: Plot of a chemical propulsion spacecraft trajectory from low Earth orbit to lunar distant retrograde orbit with a Weak Stability Boundary coast and a lunar flyby.

The corresponding data is shown in Table 1. Both chemical propulsion trajectories are much shorter trip times but also require much more propellant. This is a primary trade for spacecraft and mission architectures. The payload capability is 11,500 kg for each case.

	SEP	Chemical	WSB
Trip time (days)	1030	20	96
deltaV (m/s)	6000	3300	3200
Propellant (1000 kg)	4.1	31	29
Dry mass (1000 kg)	15	15	15

Table 1: Data for the low Earth orbit to lunar distant retrograde orbit cases.

III.II. Earth Orbit to James Webb Space Telescope Orbit

The SEP trajectory is shown in Figure 4 and the chemical propulsion trajectory is shown in Figure 5. The corresponding data is shown in Table 2.

The SEP case is almost the same trip duration as the chemical propulsion case primarily due to the orbital mechanics of this trajectory. The xenon propellant in the SEP case is considerably less than the chemical propellant case. The stay time in the JWST orbit is also less for the SEP case, again due to the orbital mechanics and differences in delta-V capability of the two

approaches. The payload capability is 23,500 kg for each case.

	SEP	Chemical
Trip time (days)	196	191
Time at JWST (days)	7	66
deltaV (m/s)	630	470
Propellant (1000 kg)	0.7	4.8
Dry mass (1000 kg)	27	27

Table 2: Data for the Earth orbit to James Webb Space Telescope orbit cases.

III.III. Earth Escape to Mars Entry

The SEP and chemical propulsion trajectories are shown in Figure 6. The corresponding data is shown in Table 3. For the Mars trip, the SEP case takes significantly longer but with much less propellant. The payload capability is 4,500 kg for each case.

One interesting footnote is the significantly lower propellant requirement for both the JWST and Mars cases compared to the lunar case. This is due to the starting point assumptions: the lunar case starting point is low Earth orbit, which is still within the Earth gravity well, and therefore still requires significant delta-V to escape. The JWST and Mars cases require similar delta-V to get to their starting points, Earth-moon L2 and Earth escape, respectively.

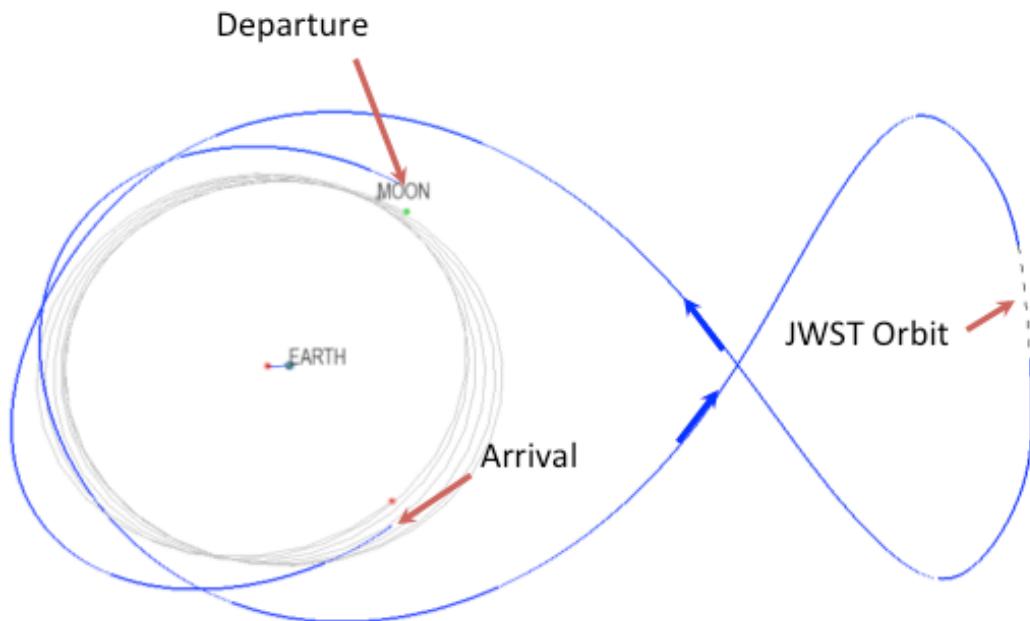


Figure 4: Plot of the SEP spacecraft trajectory to the James Webb Space Telescope orbit beginning and ending from the Earth-moon L2 Lagrangian Point. The JWST orbit is shown as a dotted line.

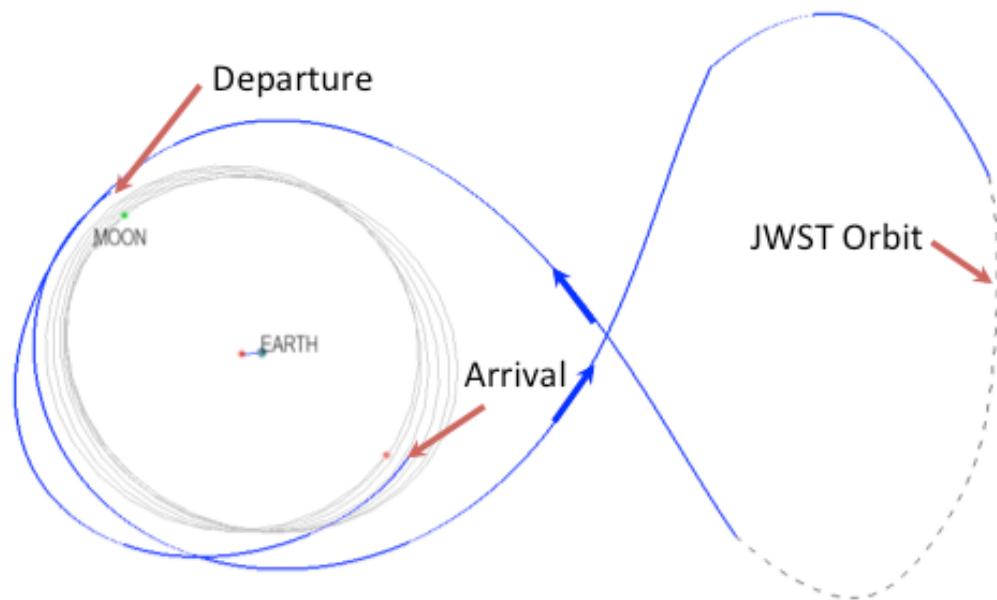


Figure 5: Plot of the chemical spacecraft trajectory to the James Webb Space Telescope orbit beginning and ending from the Earth-moon L2 Lagrangian Point. The JWST orbit is shown as a dotted line.

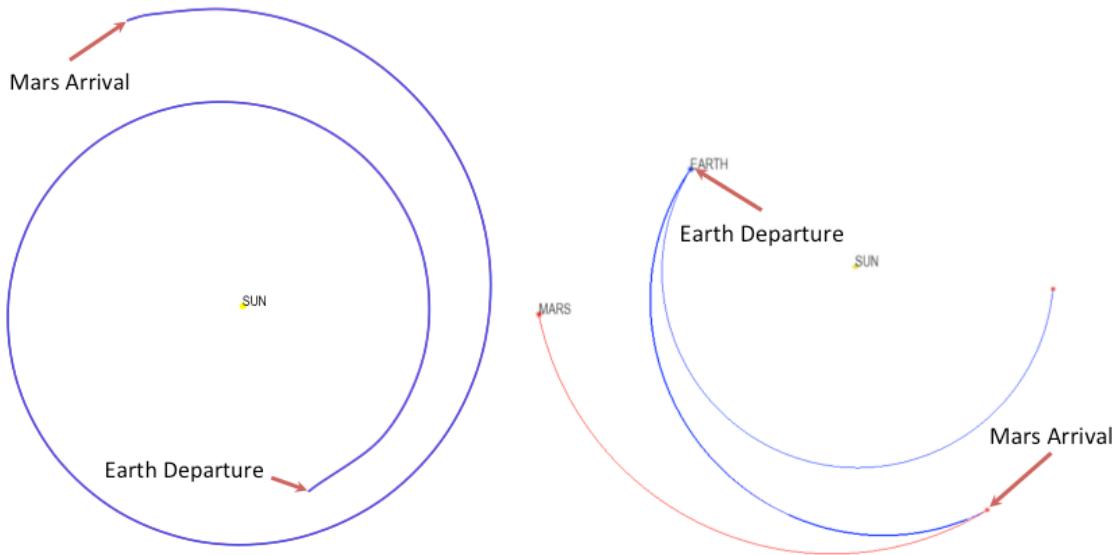


Figure 6: Trajectories from Earth escape to Mars entry. The left plot is the SEP spacecraft trajectory and the right is the chemical propulsion spacecraft trajectory.

	SEP	Chemical
Trip time (days)	730	210
deltaV (m/s)	6000	2000
Propellant (1000 kg)	1.7	7.9
Dry mass (1000 kg)	8	8

Table 3: Data for the Earth escape to Mars entry cases.

IV. TECHNOLOGY CHALLENGES

These missions highlight some of the technology challenges of the solar electric propulsion technologies. Some of the trajectories require long duration continuous thrust and also multiple on/off cycles to optimize the flight path. This thrust profile can push the boundaries of current ion thruster technology. NASA is currently focusing on more robust and longer life designs to mitigate this challenge. This includes magnetic shielding to reduce or eliminate erosion within the thruster.³ Recent testing of the 12.5 kW Hall-Effect, magnetically shielded xenon thruster is shown in Figure 7.

The long duration missions, especially in spiral trajectories around the Earth and other high radiation environments, can be problematic for radiation sensitive components. The solar array and power electronics can be protected at the expense of mass and design complexity. NASA is also focusing on mitigating these challenges as well.

NASA is also continuing to support development of higher power subsystems. This includes higher voltage power electronics and larger scale solar arrays. Challenges of larger solar arrays include in-space deployment and structural robustness of the array during launch and other high dynamic loading conditions. Two solar arrays that have been supported by NASA development include a roll-out solar array, shown in Figure 8, and a multiple-petal circular design, shown in Figure 9.

V. SUMMARY

Two propulsion subsystems are compared for their payload and transit time capabilities for three different cargo missions. For the mission to lunar orbit, the solar electric propulsion spacecraft requires much more travel time but significantly less propellant. Flying to the JWST orbit, both propulsion approaches provide similar transit times due to the orbital mechanics, but the use of SEP again significantly reduces the propellant. The chemical propulsion spacecraft does offer longer time at the JWST orbit itself. Finally, for a mission to Mars orbit, SEP requires more trip time but again saves significant propellant.

Though each of these spacecraft and trajectories for both propulsion approaches can be further optimized and iterated upon, these missions show the value of near-term solar array and electric propulsion technology and their potential in a human Mars architecture.

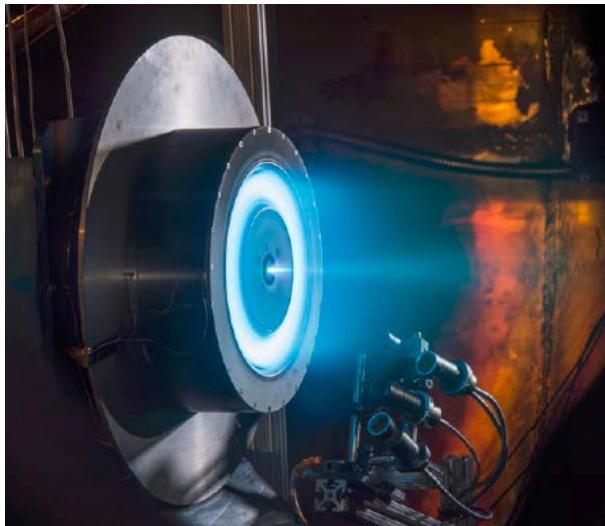


Figure 7. 12.5kW Hall-Effect Rocket with Magnetic Shielding (HERMeS) operating in VF5 at NASA GRC from Herman, et al.¹⁰



Figure 8. Engineering development unit roll-out solar array from Manzella and Hack.²



Figure 9. Engineering development unit circular solar array from Manzella and Hack.²

¹ "Pioneering Space: NASA's Next Steps on the Path to Mars," <http://www.nasa.gov/sites/default/files/files/Pioneering-space-final-052914b.pdf>, May 2014.

² Manzella, D., and Hack, K., "High-Power Solar Electric Propulsion for Future NASA Missions," 50th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Cleveland, OH, USA, July 2014.

³ Hofer, R.A. et al., "Development Approach and Status of the 12.5 kW HERMeS Hall Thruster for the Solar Electric Propulsion Technology Demonstration Mission", Joint Conference of 30th International Symposium on Space Technology and Science, 34th International Electric Propulsion Conference, and 6th Nano-satellite Symposium, Hyogo-Kobe, Japan, July, 2015.

⁴ Myers, R., "Overview of Major U.S. Industrial Electric Propulsion Programs," AIAA 2004-3331, 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Fort Lauderdale, FL, USA, July 2004.

⁵ Brophy, J. R. et al., "Development and Testing of the Dawn Ion Propulsion System," AIAA 2006-4319, 42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Sacramento, CA, USA, July 2006.

⁶ Mercer, C.R., McGuire, M.L., Oleson, S.R., Barrett, M.J., "Solar Electric Propulsion Concepts for Human Space Exploration," AIAA Space 2015, Pasadena, CA, USA, August 2015.

⁷ Burke, L.M., Martini, M.C., Oleson, S.R., "A High Power Solar Electric Propulsion - Chemical Mission for Human Exploration of Mars," 50th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Cleveland, OH, USA, July 2014.

⁸ Greenhouse, M.A., "The James Webb Space Telescope: Mission Overview and Status," AIAA 2012-5100, AIAA Space 2012, Pasadena, California, USA, September 2012.

⁹ Gates, M. et al., "NASA's Asteroid Redirect Mission Concept Development Summary," 2015 IEEE Aerospace Conference, Big Sky, MT, USA, March 2015.

¹⁰ Herman, D.A. et al., "The Ion Propulsion System for the Solar Electric Propulsion Technology Demonstration Mission", Joint Conference of 30th International Symposium on Space Technology and Science, 34th International Electric Propulsion Conference, and 6th Nano-satellite Symposium, Hyogo-Kobe, Japan, July, 2015.

¹¹ Williams, J., Senent, J.S., Ocampo, C.A., Mathur, R., "Overview and Software Architecture of the Copernicus Trajectory Design and Optimization System", 4th International Conference on Astrodynamics Tools and Techniques, May 2010.

¹²

<http://trajectory.grc.nasa.gov/tools/chebytop.shtml>